NASA TECHNICAL NOTE

NASA TN D-6342

c.1

NASA TN D-6342

USE OF ORBIT-TO-ORBIT SHUTTLES FOR HYPERBOLIC RENDEZVOUS WITH RETURNING PLANETARY SPACECRAFT

by David R. Brooks and Edwin F. Harrison

Langley Research Center
Hampton, Va. 23365

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION • WASHINGTON, D. C. • SEPTEMBER 1971

1. Report No. NASA TN D-6342	2. Government Accession No.	3. Recipient's Catalog No.
4. Title and Subtitle USE OF ORBIT-TO-ORBIT SHUTTLES FOR HYPERBOLIC RENDEZVOUS WITH RETURNING PLANETARY SPACECRAFT		5. Report Date September 1971
		6. Performing Organization Code
7. Author(s) David R. Brooks and Edwin F. Harrison		8. Performing Organization Report No. L-7664
		10. Work Unit No. 117-07-04-10
9. Performing Organization Name and Address NASA Langley Research Center Hampton, Va. 23365		11. Contract or Grant No.
		13. Type of Report and Period Covered Technical Note
12. Sponsoring Agency Name and Address National Aeronautics and Space Administration Washington, D.C. 20546		14. Sponsoring Agency Code
15. Supplementary Notes		

16. Abstract

An Earth-return mode for interplanetary spacecraft via hyperbolic rendezvous is described. In this mode an orbit-to-orbit shuttle leaves a circular Earth orbit, performs a rendezvous with a returning interplanetary spacecraft as it approaches Earth on a hyperbolic trajectory, docks and performs desired transfers, and deboosts back into a circular Earth orbit while the spacecraft continues on its hyperbolic path. Two basic flight modes are proposed, and the equations for analyzing the maneuvers are derived. The initial mass in Earth orbit required for a rendezvous system utilizing chemical or nuclear propulsion compares favorably with the initial masses in Earth orbit chargeable to transport of some conventional Earth-return systems to a target planet and back. In addition, the maneuver has considerable merit as a backup and rescue system. For typical missions, orbit-to-orbit shuttles having initial masses in Earth orbit of up to 500 000 kg are required.

17. Key Words (Suggested by Author(s)) Orbit-to-orbit shuttle; Interplanetary missions; Trajectory analysis		18. Distribution Statement Unclassified – Unlimited	
19. Security Classif. (of this report) Unclassified	20. Security Classif. (of this page) Unclassified	21. No. of Pages 32	22. Price* $3.00

*For sale by the National Technical Information Service, Springfield, Virginia 22151

USE OF ORBIT-TO-ORBIT SHUTTLES FOR HYPERBOLIC RENDEZVOUS WITH RETURNING PLANETARY SPACECRAFT

By David R. Brooks and Edwin F. Harrison
Langley Research Center

SUMMARY

An Earth-return mode for interplanetary spacecraft via hyperbolic rendezvous is described. In this mode an orbit-to-orbit shuttle leaves a circular Earth orbit, performs a rendezvous with a returning interplanetary spacecraft approaching Earth on a hyperbolic trajectory, docks and performs desired transfers, and deboosts back into a circular Earth orbit while the spacecraft continues on its hyperbolic path. Two basic flight modes are proposed, and the equations for analyzing the maneuvers are derived.

The initial mass in Earth orbit required for a rendezvous system utilizing chemical or nuclear propulsion compares favorably with the initial masses in Earth orbit chargeable to transport of a retrobraking Earth-return system to a target planet and back. The maneuver also has considerable merit as a backup and rescue system for any planetary mission even when not suitable as a primary recovery mode. For a typical projected mission (manned Mars), it is shown that orbit-to-orbit shuttles require initial masses in Earth orbit up to 500 000 kg, depending on the specific impulse of the propulsion system and whether or not the propellant tanks are discarded. The possible effect of long thrust times, resulting in gravity losses, is examined and found to be negligible for the flight modes considered. It is shown to be possible to perform large plane changes with relatively small velocity increments by taking advantage of the low spacecraft velocities at the apoapses of the required ellipses.

INTRODUCTION

The problem of Earth return following interplanetary flight has been a primary topic of study for mission planners since the inception of the space program (refs. 1 to 5). For the purposes of this study, the term "Earth return" implies return of a spacecraft to a low circular Earth orbit and does not include return to the Earth's surface from orbit. One of the most obvious methods of returning to Earth orbit is to remove excess energy by onboard propulsive means. However, limited launch capability and space propulsion by current chemical systems have made such a straightforward operational mode generally unattractive because transporting propellant for retrobraking rockets to a target planet and back imposes unacceptable weight penalties. The technology for direct return of Apollo spacecraft to the Earth's surface is not immediately applicable to many projected interplanetary missions, because the large return velocities prohibit aerobraking either to orbit or directly to the Earth's surface with present systems. A choice must be made among competing alternatives: further improvement of aerodynamic-reentry technology, acceptance of the weight penalties associated even with advanced onboard propulsion for retrobraking, or development of new approaches to Earth-return operations and systems based on projected technology which should be available by the time round-trip planetary missions are undertaken.

In this technical note, investigation is made of an approach based on projected technology involving an orbitally based orbit-to-orbit shuttle (OOS) (ref. 6) which has the propulsive capability for interception of, rendezvous with, and return of the payload from interplanetary spacecraft as they approach Earth on a hyperbolic trajectory. The advantage of hyperbolic rendezvous as a primary recovery mode is immediately clear: The returning interplanetary spacecraft need not have carried its own Earth-return system all the way to the destination and back. The feasibility of using such a system as a primary Earth-return mode for planetary missions is examined in this technical note. It is clear that regardless of the results of comparisons between this and other return modes, hyperbolic rendezvous has considerable merit as a backup and/or rescue mode in the event of failure of the primary Earth-return system (e.g., loss of propellant).

Hyperbolic rendezvous is not a new concept (ref. 7), but its applicability has always been restricted by the relatively primitive state of propulsion systems and space operational capabilities. It will be made feasible only by the advent of a new era in space transportation (ref. 8) dominated by a reusable Earth-to-orbit shuttle with vastly increased flexibility for Earth orbital maneuvering. Such a system invites consideration of operations which can be made feasible and even desirable by the availability of low-cost launch capability and sophisticated orbital facilities. It should be emphasized at the outset that for the purposes of this study, the OOS and associated reusable launch systems required for support of the hyperbolic rendezvous mission are assumed to be available and totally operational as components of an advanced space transportation system. The development of such systems for the sole purpose of serving as a primary or alternate Earth-return mode for manned space flight could not be justified. The operation considered in this study should be combined with others to justify construction of space transportation systems and to assure their maximum utilization once they are operational.

The present study is divided into two parts. In the first, the problem of hyperbolic rendezvous is examined in a general way by developing appropriate equations from basic principles. In the second part, Earth return via hyperbolic rendezvous is conceptually applied to representative interplanetary missions and compared with alternate Earth-return modes.

SYMBOLS

a	semimajor axis, km
$A = \exp\left(\frac{\Delta V}{I_{sp} g}\right)$	
e	eccentricity, dimensionless
g	Earth sea-level gravitational acceleration, 0.0098 km/sec^2
i	orbital inclination with respect to OOS starting orbit, deg
I_{sp}	specific impulse, sec
m	propellant mass, kg (see appendix B)
m_1, m_2, m_3, m_4	propellant masses required for the four main maneuvers of a manned Mars mission, kg (see appendix C)
M	mass of a rocket stage, kg
M_o	initial total mass in Earth orbit, kg
p	semilatus rectum, km
P	point on a hyperbolic trajectory at which rendezvous is assumed to take place (see also under subscripts)
r	radius, km
t	time, min
T	thrust, newtons (N)
V	velocity, km/sec
ΔV	velocity increment, km/sec (total one-way increment: $\Delta V_{total} = \Delta V_1 + \Delta V_2$)
α	fraction of propellant mass added for tanks, dimensionless
γ	flight-path angle, deg

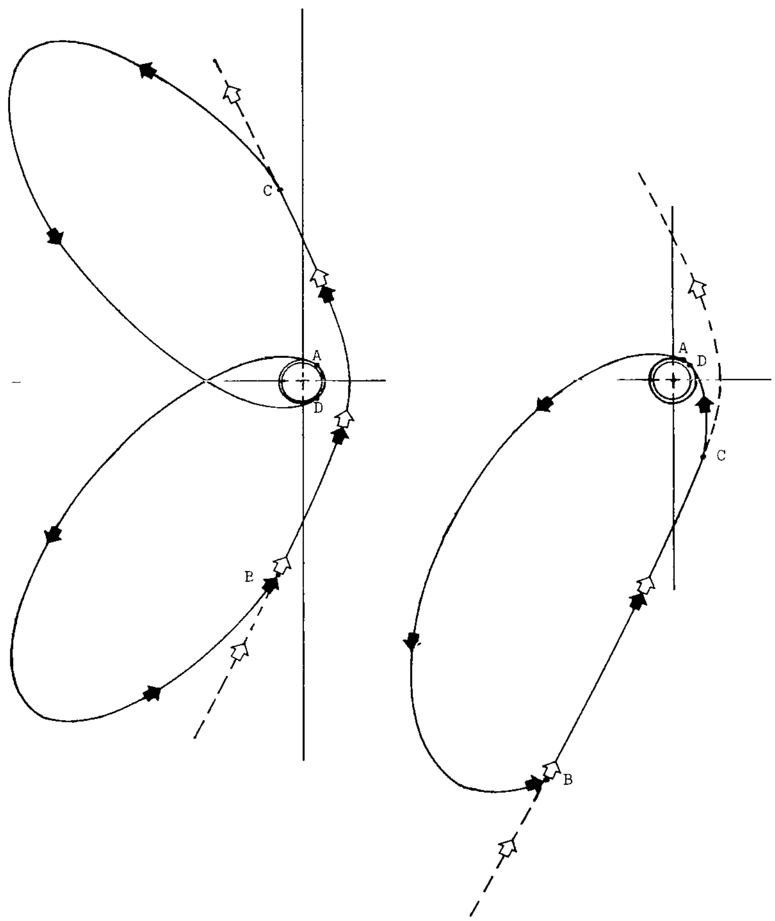
$\Delta\gamma$	difference between elliptic and hyperbolic flight-path angles, deg
θ	true anomaly, deg
Θ	angle between elliptic axis and hyperbolic axis, deg
μ	Earth's gravitational constant, 3.9858×10^5 km^3/sec^2

Subscripts:

A	apoapsis
E	elliptic
EN	Earth entry; when applied to velocity, the velocity of a spacecraft at an altitude of 122 km
H	hyperbolic
i,j	summation indicator representing the final burn in a multiburn sequence
max	maximum
min	minimum
n	indicates quantity associated with nth burn of a multiburn sequence
P	periapsis
R	rendezvous point
t	tangential
1	maneuver from Earth orbit
2	maneuver at the rendezvous point
$\oplus$	Earth
∞	infinity; when applied to velocity, indicates hyperbolic excess velocity

DESCRIPTION OF HYPERBOLIC RENDEZVOUS MANEUVERS

The basic assumed paths for a round-trip hyperbolic rendezvous with a returning spacecraft are shown in figure 1. The two modes illustrated are symmetrical and unsymmetrical with respect to the hyperbolic periapsis. The solid arrows indicate possible trajectories for the orbit-to-orbit shuttle; the open arrows indicate the spacecraft path along a hyperbola.



(a) Symmetrical mode. (b) Unsymmetrical mode.

Figure 1.- Two possible four-impulse coplanar rendezvous and return mission modes. The four impulses are A - leave circular Earth orbit, B - match hyperbolic trajectory, C - deboost to return trajectory, D - reenter circular Earth orbit.

Symmetrical Rendezvous Maneuver

In figure 1(a), an orbit-to-orbit shuttle (OOS) leaves a circular Earth orbit at point A after determining the proper transfer ellipse for rendezvous with the spacecraft at a point B on the incoming hyperbolic trajectory. The point B is selected as a result of a compromise among competing demands imposed by the desire to minimize time on the transfer ellipse, while at the same time minimizing propulsion requirements and allowing sufficient time from periapsis of the hyperbola. At B, the OOS performs the final rendezvous and docking maneuver. Transfers of crew, modules, or samples are performed on the hyperbolic trajectory between B and C. At C, the remaining portion of the interplanetary spacecraft continues on the hyperbola, while the OOS deboosts into the return ellipse with its cargo. The symmetrical mode provides a relatively long transit time on the return trajectory (possibly several days). Thus, the parameters of the transfer ellipse may be easily adjusted for proper time phasing to bring the OOS into position for returning to a circular orbit at D near a space station or some other site chosen for debriefing and quarantine, in the case of a manned mission, or simply isolation of the cargo, in the case of an unmanned sample return.

Unsymmetrical Rendezvous Mode

Figure 1(b) illustrates an unsymmetrical mode in which the rendezvous (point B) and deboost (point C) maneuvers are both performed prior to passing the hyperbolic periapsis. Trajectory characteristics for a typical example of both modes will be treated quantitatively in a later section. Qualitatively, the symmetrical mode (fig. 1(a)) might have the advantage of making timing and orbit phasing a little easier, with the possible disadvantage of very long mission times (many days). The unsymmetrical mode (fig. 1(b)) might reduce the total mission time by making the return trip much shorter, but this may be difficult to achieve from the standpoint of propulsion and guidance. Also, whereas a plane change requires only a small velocity increment at the apoapsis of large transfer ellipses, a much larger velocity increment would be needed to make a plane change onthe short return leg of the rendezvous mode shown in figure 1(b).

The unsymmetrical mode can also be carried out with rendezvous after the hyperbolic periapsis is passed and with the long elliptical portion of the trajectory as the deboost phase. If two identical OOS assemblies were available, it would be possible, in principle, to use the rendezvous after periapsis as a backup mode to a rendezvous and deboost before periapsis.

A special case of the second mode will occur if the periapsis of the hyperbola is coincident with the altitude of the starting orbit. Then, the maneuvers at C and D would be combined into one large velocity change.

EQUATIONS FOR HYPERBOLIC RENDEZVOUS

In appendix A, the general equations for hyperbolic rendezvous are derived from the basic conic equations found in the NASA Orbital Flight Handbook (ref. 9). From these equations, limits are established on the allowable transfer ellipses so that it is possible to compute the velocity changes necessary to perform the rendezvous maneuver, taking into account additional constraints which will be discussed in a later section.

The geometry for the problem is defined in figure 2. For deriving the equations it is assumed that the OOS (solid arrows) will leave a circular Earth orbit and meet the returning spacecraft (open arrows) on the incoming hyperbolic trajectory at point P'. The circular Earth orbit and the hyperbolic return trajectory are assumed to be coplanar, with the OOS always available at the right place and time in Earth orbit; that is, the problems of phasing are ignored. It is obvious from figure 2 that the same size transfer ellipse can also intersect the hyperbola at point P. The same equations result in either case, but the transit time from hyperbolic periapsis to P' is positive and the time from hyperbolic periapsis to P is negative.

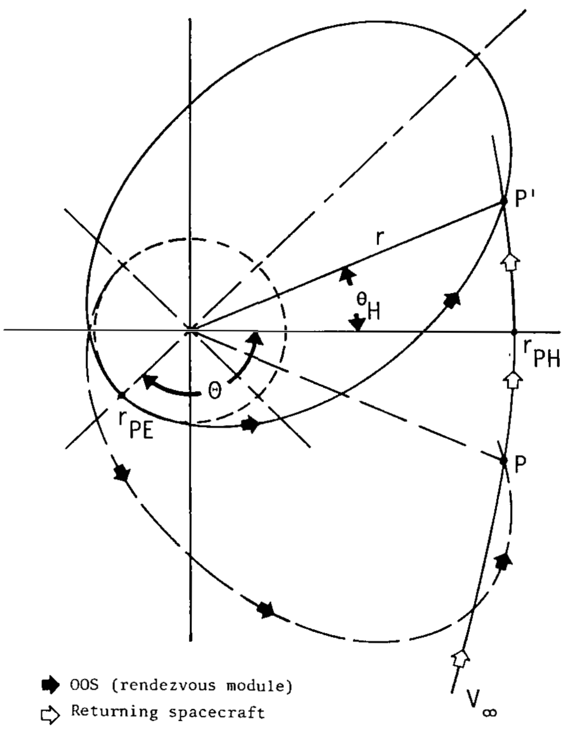


Figure 2.- Geometry for derivation of hyperbolic-rendezvous equations.

The geometry of the circular Earth orbit is completely specified by its radius, which has the same value as the periapsis of the transfer ellipse r_{PE}. The hyperbolic return trajectory is specified in terms of quantities appropriate to its consideration as a planetary return, that is, the distance to the hyperbolic periapsis r_{PH} and the hyperbolic excess velocity V_∞.

In appendix A, it is shown that the ellipses for transfer between Earth orbit and P may have the following range of eccentricities:

$$\frac{r - r_{PE}}{r + r_{PE}} \leqq e_E < 1 \tag{1}$$

where r is the distance to the rendezvous point. Other appropriate relationships derived in appendix A show the angular orientation of a rendezvous maneuver which would involve only a tangential burn at the intersection point.

For each set of initial conditions, that is, values of r_{PE}, r_{PH}, and V_∞, it is necessary to compute the velocity changes required to perform the rendezvous for the range of transfer ellipses specified by expression (1). The rendezvous maneuver is assumed to consist of two impulsive velocity changes: ΔV_1 removes the rendezvous vehicle from circular orbit and places it on the transfer ellipse, and ΔV_2 matches the velocity vector of the rendezvous vehicle with the return vehicle. The elliptical-periapsis velocity minus the circular velocity is ΔV_1, and ΔV_2 is found by the law of cosines:

$$\Delta V_1 = \sqrt{\mu\left(\frac{2}{r_{PE}} - \frac{1}{a_E}\right)} - \sqrt{\mu\left(\frac{1}{r_{PE}}\right)} \tag{2}$$

$$\Delta V_2 = \sqrt{V_E^2(P) + V_H^2(P) - 2V_E(P)V_H(P)\cos\left[\Delta\gamma(P)\right]} \tag{3}$$

where

$$\Delta\gamma(P) = \gamma_H(P) - \gamma_E(P) \tag{4}$$

In general it would be desirable to search analytically for the conditions which minimize the sum of ΔV_1 and ΔV_2, but there are at least two constraints on Earth-orbital application of hyperbolic rendezvous which make any such exercise unnecessary. First, if reasonable values for starting-orbit altitudes and hyperbolic excess velocities are assumed, the time spent on the transfer ellipse can become excessive for the lowest ΔV transfers. Second, the minimum time required for performing maneuvers after rendezvous, such as docking, crew transfer, or sample stowage, puts limits on transfers close to the hyperbolic periapsis; the ellipses excluded for these reasons may include those with the lowest total ΔV requirement.

The first constraint is illustrated by figure 3, which shows time spent on the transfer ellipse as a function of total one-way velocity increment for a hyperbola having $V_\infty = 6.1$ km/sec and $r_{PH} = 12\ 756$ km, or $r_{PH}/r_\oplus = 2$. The transfer ellipse is

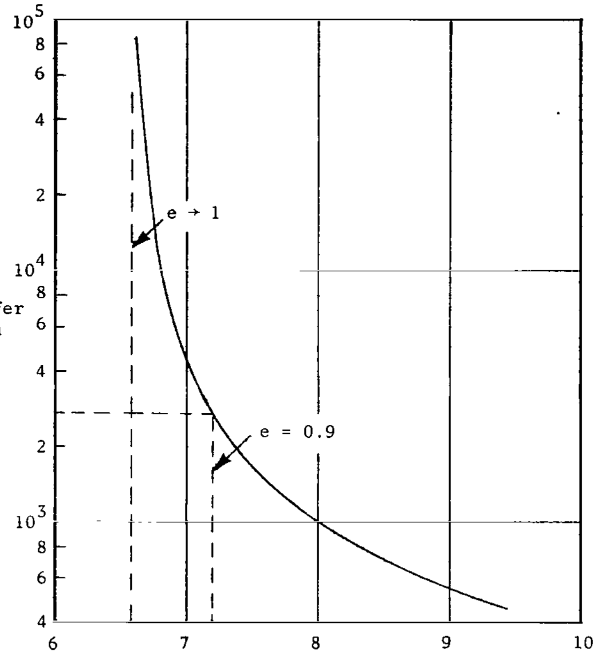


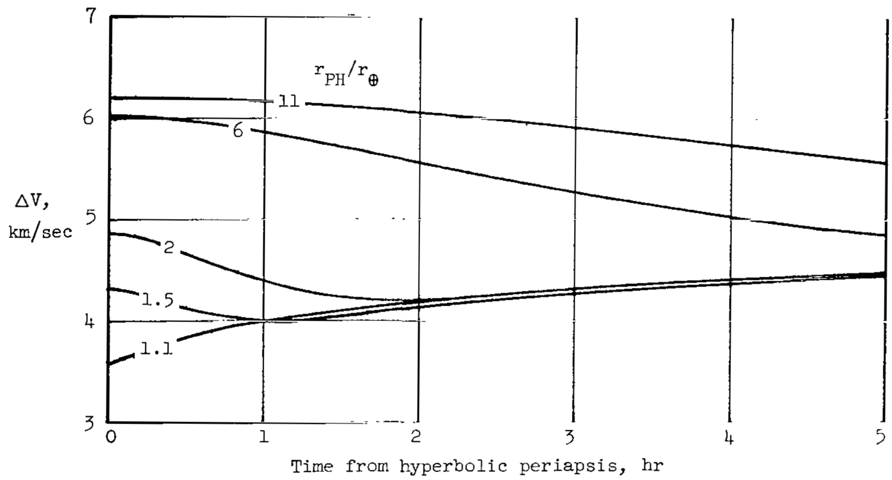
Figure 3.- Time spent on transfer ellipse as a function of total one-way velocity increment. $V_\infty = 6.1$ km/sec; $t_{PH} = 2$ hr; $r_{PH}/r_\oplus = 2$.

assumed to start from a 500-km circular orbit[1] $(r_{PE} = 6878\text{ km})$, and the assumed rendezvous point is 2 hours before passing the hyperbolic periapsis. The minimum ΔV for this case occurs as the eccentricity of the transfer ellipse approaches 1, that is, as the time spent on the ellipse approaches infinity. Restricting the size of the transfer ellipse leads to increases in ΔV. For example, with the conditions given in figure 3, the ΔV for an ellipse having $e_E = 0.9$ is about 9 percent greater (7.2 km/sec) than the minimum (6.6 km/sec), and 46 hours and 12 minutes are required on the transfer ellipse.

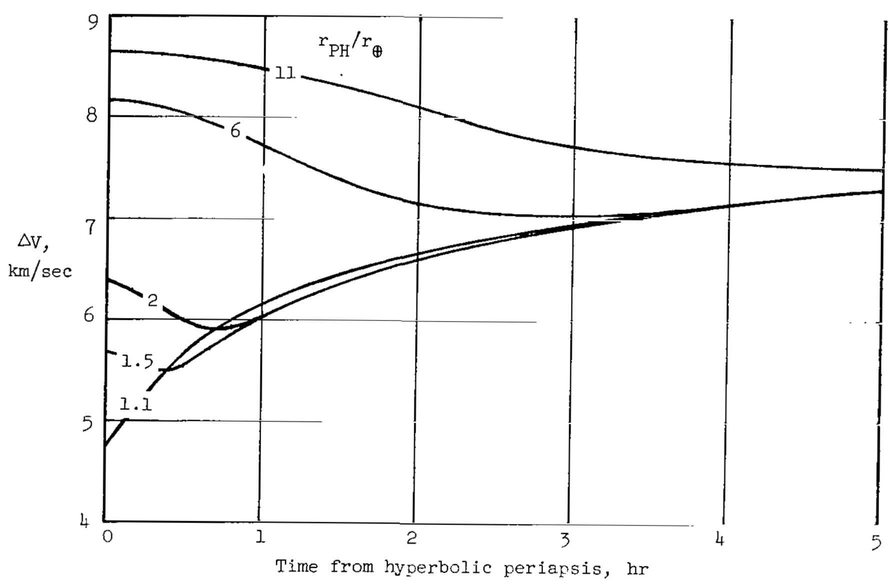
The second constraint, involving the trade-off between ΔV and time from hyperbolic periapsis, is illustrated in figure 4. The minimum one-way velocity increment (which may require an infinitely large transfer ellipse) is shown as a function of time

[1]Nominal altitude for the proposed space station.

from hyperbolic periapsis for the hyperbolic excess velocities of 3.05, 6.10, and 9.15 km/sec in figure 4(a), 4(b), and 4(c), respectively. For example, for V_∞ = 6.10 km/sec, figure 4(b) shows that the minimum ΔV occurs at about 40 minutes from hyperbolic periapsis when $r_{PH}/r_\oplus = 2$. Figure 4 shows that especially for larger values of time from periapsis, the ΔV is not a strong function of the hyperbolic periapsis distance. This fact could be important in the case of a rescue operation involving a disabled spacecraft with limited capabilities for maneuvering into a particular hyperbolic

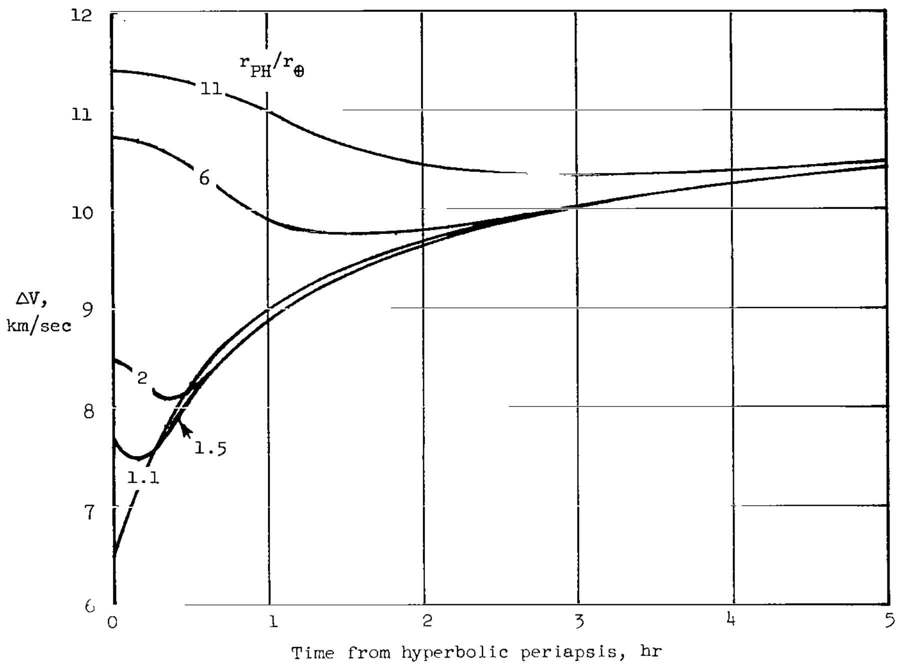


(a) V_∞ = 3.05 km/sec.



(b) V_∞ = 6.10 km/sec.

Figure 4.- Minimum one-way ΔV as a function of time from hyperbolic periapsis for five values of the ratio of hyperbolic periapsis radius r_{PH} to Earth radius $r_\oplus$.



(c) V_∞ = 9.15 km/sec.

Figure 4.- Concluded.

trajectory – the lack of sensitivity to closest approach distance allows tracking and maneuvering to be done by the rendezvous vehicle or by ground stations, with the returning spacecraft passive during the operation.

APPLICATION OF HYPERBOLIC RENDEZVOUS TO INTERPLANETARY MISSIONS

The primary usefulness of a hyperbolic-rendezvous maneuver lies in its application in either a nominal or emergency mode to returning interplanetary missions. Hyperbolic excess and Earth-entry velocities for representative planetary missions are shown in figure 5, which is from reference 10; V_∞ ranges from 5 to 18 km/sec and the corresponding reentry velocities are as high as 21 km/sec. For the present study initial mass in Earth orbit M_0 will be used to compare hyperbolic rendezvous with other Earth-return modes, but it should be emphasized that the relative desirability of return modes depends on other factors also, such as system reliability and cost of developing atmospheric-entry systems to cope with the expected high return velocities.

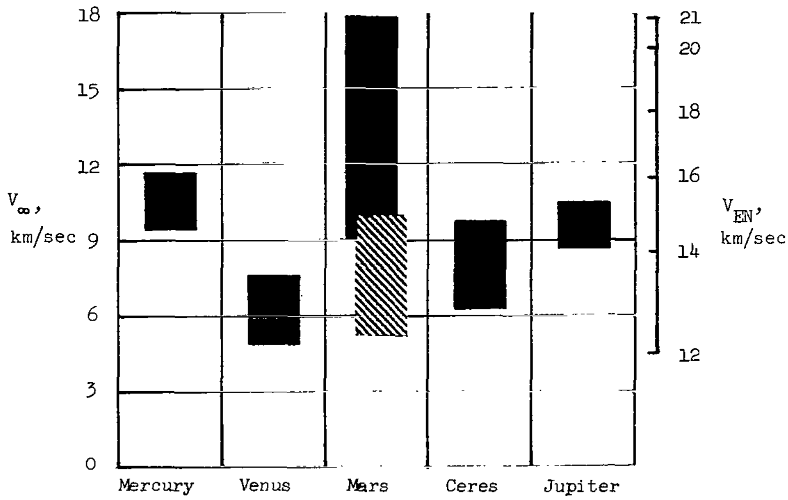


Figure 5.- Earth-entry and hyperbolic excess velocities for representative planetary missions. Hatched area indicates Venus swingby.

Vehicle and Trajectory Constraints and Assumptions

For the purpose of examining hyperbolic rendezvous as applied to planetary missions, several assumptions must be made concerning, first, elements of the hardware involved, and second, some details of the proposed trajectories to be followed by both the returning spacecraft and the OOS. The allowable trajectories must result in reasonable total mission times and must require values of ΔV obtainable with existing or planned chemical or nuclear propulsion systems. The OOS is assumed to consist of four parts: a rendezvous module, a fixed inert mass, tanks, and propellant. Propulsion is either chemical or nuclear.

Rendezvous module.- The rendezvous module houses the crew and contains all necessary systems for life support, guidance, communications, and so forth. Its mass is assumed to be between 2700 and 4500 kg and to be independent of the type of propulsion used or the ΔV required for a particular mission. For the return trip it has been assumed that an additional 900 kg of cargo, corresponding to a six-man crew and samples, for example, is onboard.

Chemical propulsion system.- A typical proposed OOS configuration for low-energy missions has the following characteristics (unpublished data from Aerospace Corporation):

I_{sp}, sec	460
M_O, kg	31 300
Propellant, kg	27 700
Tank and tank-related structure (micrometeoroid shielding and thermal insulation), kg	1500
Inert mass, kg	2100

For such a system the tank fraction α is 5.4 percent and this value has been assumed for the study. The inert mass is taken to be constant even though a much different configuration may be required for very high-energy missions. For a small OOS, RL10 (or similar) engines with a thrust of about 67 000 N per engine are envisioned, and their mass is assumed to be included in the 2100 kg. For a larger stage, a J-2 class engine, with a thrust of about 900 000 N is envisioned, but the inert mass is still fixed at 2100 kg. This simplification is justified by the uncertainty allowed for the rendezvous module and further by the realization that so much propellant is required for a typical hyperbolic-rendezvous mission that the mass of the tanks may be more than that of the inert stage.

Nuclear propulsion system.- A solid-core hydrogen-fueled reactor is assumed to provide a specific impulse I_{sp} of 850 seconds with a thrust of 900 000 N. The total inert-stage mass, including the engine and shielding (but not tanks) is 13 600 kg. The assumed parameters correspond to a large NERVA configuration (ref. 11). Tanks are assumed to weigh 15 percent of the propellant – 13 percent for structure and 2 percent for contingency (ref. 12). A typical nuclear OOS proposed for Earth-Moon shuttle service has an initial mass in Earth orbit of 180 000 kg (unpublished data from North American Rockwell Corporation).

Mission times.- There are two timing problems which must be settled before applying hyperbolic rendezvous to a hypothetical mission. First, the total mission time should be limited in some way, and second, the time on the hyperbola must be sufficiently long to allow whatever maneuvers are required. For the first problem, it is convenient to limit the transfer ellipses to 50 hours. (For a periapsis of 6878 km this corresponds almost exactly to an eccentricity of 0.9.) Such a limit restricts round-trip times to a few days without substantially increasing the ΔV requirements. (Recall fig. 3.)

The problem of allowing sufficient time on the hyperbola for maneuvers and transfer procedures has been approached by referring to experience gained from the Apollo 11 and 12 missions during the docking of the lunar module (LM) and the command service module (CSM) following lunar lift-off. Table 1 (ref. 13 and unpublished data from Manned Spacecraft Center) shows a time history of the CSM/LM docking procedures for the Apollo 11 and 12 missions, starting from initiation of rendezvous radar tracking and ending with the CSM/LM separation maneuver. These times are considered to be representative of the times required in the case under study. For Apollo, about $3\frac{1}{2}$ hours were required for rendezvous, docking, and transfer of two men, their equipment, and samples. For representative proposed planetary missions, up to six men are involved; therefore, it is assumed that a total time somewhat longer than for Apollo, or about 4 hours, should be allowed for rendezvous, docking, and transfer of crew and/or cargo. For the symmetrical mission mode of figure 1(a), this corresponds to a rendezvous at 2 hours before periapsis. Since the allowed time can only be considered as a rough guess at best, it is

TABLE 1.- MISSION TIMES FOR RENDEZVOUS MANEUVER BETWEEN LM AND CSM FOLLOWING LUNAR LIFT-OFF OF APOLLO 11 AND 12

Maneuver	Apollo 11		Apollo 12	
	Mission time, hr:min	Cumulative elapsed time, hr:min	Mission time, hr:min	Cumulative elapsed time, hr:min
Terminal-phase initiation	127:04	---	144:36	---
Terminal-phase finalization	127:46	0:42	145:06	0:30
Docking completed	128:03	0:59	145:36	1:00
LM/CSM separation	130:10	3:06	148:00	3:24
Final separation maneuver completed	130:30	3:26	148:05	3:29

fortunate that the velocity requirements are only weakly sensitive to an increase in the time – in fact, the velocity increments for longer times from periapsis actually decrease for some values of r_{PH}. (See fig. 4.)

Spacecraft trajectory.- Since it has been shown previously that in general, ΔV is relatively insensitive to r_{PH}, it is only necessary to select a representative value of r_{PH} to use in ΔV computations for a particular case. For this purpose $r_{PH}/r_{\oplus} = 2$ will be the nominal value. Also, for the nominal trajectories, the spacecraft trajectory is assumed to be coplanar with the rendezvous vehicle. (Out-of-plane maneuvers will be discussed briefly in a separate section.)

Trajectory Data for Hyperbolic Rendezvous Maneuvers With Interplanetary Spacecraft

Trajectory data for a hyperbolic rendezvous maneuver have been computed for several values of V_∞ up to 12.19 km/sec (40 000 ft/sec) from the equations of appendix A. The symmetric mode of figure 1(a) has been chosen, and ΔV required for an entire mission is therefore $2 \times \Delta V_{total}$. These data are shown in table 2. For each value of hyperbolic excess velocity V_∞, table 2 includes the eccentricity of the spacecraft hyperbola e_H, the magnitude of the difference between hyperbolic flight-path angles at rendezvous $|\Delta\gamma_R|$, the rendezvous radius r_R, velocities on the ellipse and hyperbola at rendezvous V_{ER} and V_{HR}, the velocity increment required for the rendezvous maneuver ΔV_2, the total one-way velocity increment ΔV_{total}, and the time spent on the transfer ellipse t_R. Spacecraft trajectory assumptions have been utilized, as discussed in the preceding section, and the starting orbit is circular, with an altitude of 500 km $\left(r_{PE} = 6878 \text{ km}\right)$ as previously assumed. For a 50-hour ellipse $\left(e_E = 0.9\right)$ starting from this altitude, $\Delta V_1 = 2.88$ km/sec (appendix A or eq. (2)).

TABLE 2.- CALCULATED TRAJECTORY DATA FOR THE

HYPERBOLIC RENDEZVOUS MANEUVER

$$\left[\begin{array}{l} r_{PH}/r_{\oplus} = 2 \\ t_{PH} - t_R = 2 \text{ hr} \\ e_E = 0.9 \text{ (50-hr period)} \\ \Delta V_1 = 2.88 \text{ km/sec} \end{array}\right]$$

V_∞, km/sec	e_H	r_R, km	$\lvert\Delta\gamma_R\rvert$, deg	V_{ER}, km/sec	V_{HR}, km/sec	ΔV_2, km/sec	ΔV_{total}, km/sec	t_R, hr:min
1.52	1.074	38 150	4.82	3.89	4.82	1.00	3.88	47:51
3.05	1.297	42 130	.65	3.62	5.31	1.69	4.57	47:31
6.10	2.188	55 690	7.79	2.92	7.17	4.30	7.18	46:12
9.14	3.673	73 430	13.44	2.25	9.72	7.55	10.43	45:01
12.19	5.752	93 080	18.66	1.66	12.54	10.97	13.85	40.54

A sketch of a representative trajectory based on the data of table 2 is shown in figure 6 for $V_\infty = 6.1$ km/sec. The distances involved are normalized in terms of the Earth's radius. The ellipse has a periapsis of 6878 km and an apoapsis of about 131 000 km. The rendezvous radius is 55 690 km.

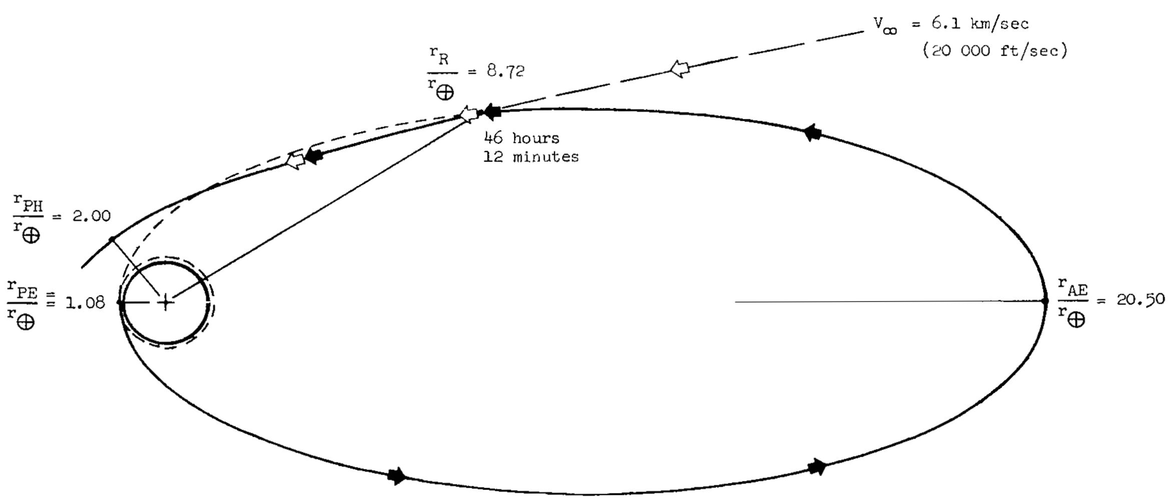


Figure 6.- Representative transfer ellipse for hyperbolic rendezvous, based on the data of table 2. V_∞ = 6.1 km/sec.

It was suggested in the Introduction of this paper that the unsymmetrical rendezvous mode of figure 1(b) might result in a smaller value of ΔV for the total mission than the symmetrical mode. As an example, consider the case for $V_\infty = 6.1$ km/sec, $r_{PH}/r_\oplus = 2$, and $t_{PH} = 2$ hours. The symmetrical mode, with 50-hour ellipses, requires a total ΔV for the round trip of $2 \times 7.18 = 14.36$ km/sec (from table 2). Figure 3 shows that by using infinitely large ellipses, this can be reduced to an absolute minimum round-trip requirement for ΔV of $2 \times 6.60 = 13.20$ km/sec. The unsymmetrical mode, as illustrated in figure 1(b), could use, as a limiting case, an infinite ellipse for rendezvous and a parabolic segment at the deboost point. As shown in figure 4(b), the minimum total one-way velocity increment for the deboost occurs at about 40 minutes before periapsis and is equal to 5.90 km/sec. To maintain a 4-hour allowance for crew transfer, the OOS must intersect the hyperbola at 4 hours and 40 minutes before periapsis. The minimum total one-way velocity increment for this maneuver is about 7.25 km/sec. Therefore, the total value of ΔV for the mission could be as low as 13.15 km/sec. If the rendezvous transfer ellipse is restricted to 50 hours in the unsymmetrical case, ΔV for the mission rises to about 14.6 km/sec. Thus, it can be seen that the difference between these two modes is not significant with respect to velocity requirements, and the choice would have to be made on the basis of other operational considerations.

Comparison of Hyperbolic Rendezvous With Conventional Earth-Return Modes

With the constraints and assumptions discussed in the previous section, the initial mass in Earth orbit has been computed as a function of hyperbolic excess velocity for two operational modes: (a) propellant tanks retained for reuse, and (b) propellant tanks expended after each of the first three engine burns. A symmetrical rendezvous and deboost maneuver is assumed, as in figure 1(a). The equations for these computations are detailed in appendix B and the results are summarized in figure 7, which shows M_0 plotted as a function of V_∞ and V_{EN} for expendable and reusable tanks and for both chemical and nuclear propulsion. The upper and lower boundaries on each curve correspond to 4500- and 2700-kg rendezvous modules, respectively.

To provide a comparison with the requirements of a typical class of planetary missions, the rectangular hatched areas in figure 7 give V_∞ and M_0 limits for some manned Mars missions which utilize a Venus swingby on the outbound or return trip. The data on which the hatched areas are based are given in table 3. An explanation of the assumptions and procedures for generating table 3 are given in appendix C, along with a sample mission worked out in detail. In figure 7, the Mars data refer to the M_0 required to provide for the transport of aerobraking (bottom area) or retrobraking (top area) Earth-return capability for a six-man crew to Mars and back. Thus, the comparison

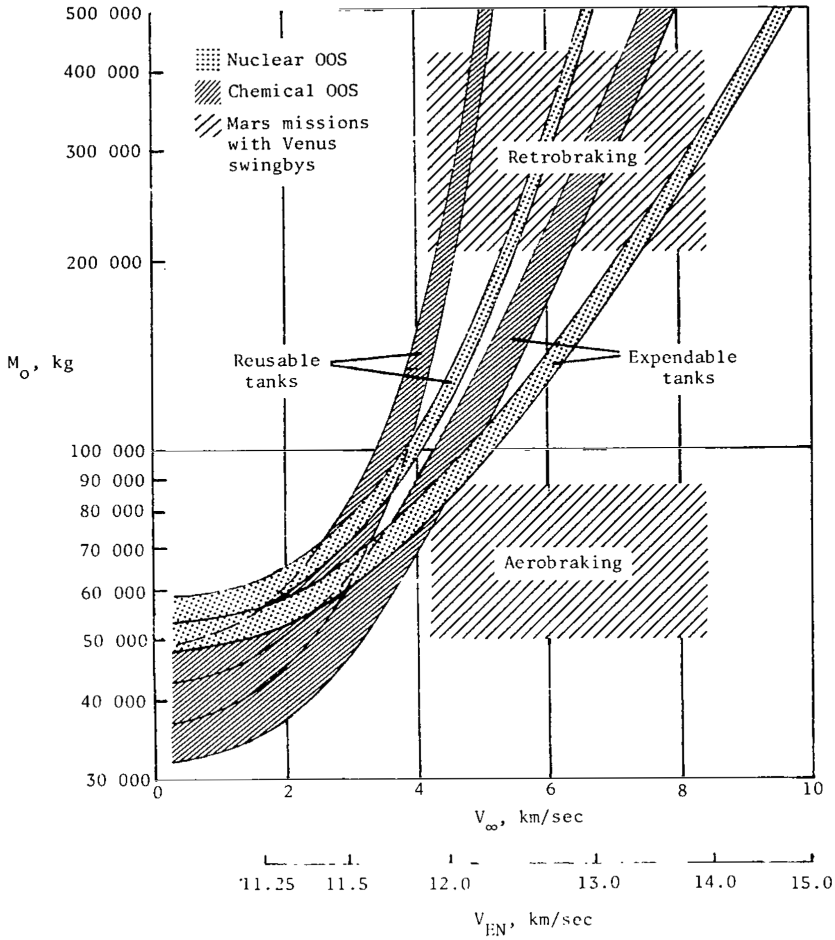


Figure 7. - Initial mass in Earth orbit as a function of hyperbolic excess velocity and Earth entry velocity, compared with manned Mars missions utilizing aerobraking or retrobraking for Earth return (appendix C).

to be made is between this mass and the M_0 of the hyperbolic-rendezvous system. As noted in appendix C, the aerobraking mode includes an 8200-kg reentry capsule, and retrobraking assumes a 2700-kg crew module which deboosts to a 500-km circular Earth orbit. For hyperbolic rendezvous no special return module would be provided.

Clearly, the aerobraked Earth return represents the smaller M_0 of the two conventional systems. The M_0 chargeable to the Earth-return system consists of the aerobraking capsule itself, a small amount of propellant for maneuvering the capsule upon return to the vicinity of the Earth, and propellant to transport that weight to Mars and back. The M_0 chargeable to the retrobraked system includes the mass of the crew module, propellant for the retrobraking maneuvers, and propellant for transporting the entire retrobraking system to Mars and back.

TABLE 3.- INITIAL MASS IN EARTH ORBIT REQUIRED FOR VARIOUS MARS MISSION OPPORTUNITIES AND EARTH-RETURN OPTIONS

Year	Venus swingby	M_0, kg, for -			ΔM_0 [b], kg, for -		V_∞ at Earth return, km/sec
		Rendezvous[a]	Aerobrake	Retrobrake	Aerobrake	Retrobrake	
1982	Inbound	548 000	602 000	793 000	54 000	245 000	6.07
84	Inbound	671 000	746 000	1 100 000	75 000	429 000	8.28
86	Outbound	574 000	628 000	817 000	54 000	243 000	6.04
88	Inbound	502 000	552 000	773 000	50 000	271 000	7.68
90	Outbound	797 000	885 000	1 206 000	88 000	409 000	6.37
93	Outbound	612 000	665 000	819 000	53 000	207 000	4.20
95	Inbound	549 000	603 000	862 000	54 000	313 000	8.43
99	Outbound	660 000	723 000	941 000	63 000	281 000	5.89

[a]Does not include the OOS.

[b]ΔM_0 is the additional M_0 required for transporting an aerobraking or retrobraking system to Mars and back when hyperbolic rendezvous is not used.

Figure 7 illustrates that hyperbolic rendezvous with either reusable or expendable tanks can be competitive on a weight basis with other primary Earth-return modes for some planetary missions. The values of V_∞ for the Mars missions utilizing Venus swingby are representative of those encountered in other planetary missions, as shown in figure 5, and the masses in Earth orbit given in figure 7 for retrobraking and aerobraking Earth-return systems are representative of those expected for other planets as well. Therefore, the applicability of figure 7 is not restricted to the Mars missions shown.

The chemical systems required for planetary return missions would be considerably larger than those presently contemplated for "space tug" applications (ref. 6) and, hence, would probably not be available. However, their smaller inert mass gives them an advantage over the nuclear-engine assemblies for missions involving small hyperbolic excess velocities. For the nuclear systems, the maximum M_0 required to meet the demands of the most energetic Mars return shown in figure 7 is about 430 000 kg and is not an unreasonable growth capability to impose on a vehicle capable of Earth-Moon shuttle service.

Incorporation of the systems indicated by figure 7 into the space transportation plan as currently envisioned is an important problem area lying outside the scope of this study. However, the sophistication required for large-scale orbital operations, including hyperbolic rendezvous, is no greater than that required for assembly in orbit of a large spacecraft for, as in the examples considered in this study, a manned Mars mission, which may require an initial mass in Earth orbit as large as a million kilograms. In the opinion of the authors, a manned round-trip planetary mission would not be undertaken without first achieving operational status for the space transportation system. Hence, hyperbolic rendezvous should be considered a contender for use as a primary recovery mode.

Even if aerobraking or onboard retrobraking are used as primary Earth-return modes for future planetary missions, the backup and rescue capability provided by application of the orbit-to-orbit shuttle can result in an increase in mission safety and probability of success with only modest effort to insure compatability between the spacecraft and the shuttle – a compatability which may be required in any event. It may also be possible to back up the hyperbolic-rendezvous system itself by using an unsymmetrical rendezvous mode and deboost before periapsis. A second OOS could be standing by to perform a mirror image of the planned maneuver after periapsis in case the first OOS fails. However, availability of such redundancy, considering the size of the systems involved, presupposes a massive orbital capability which lies far in the future.

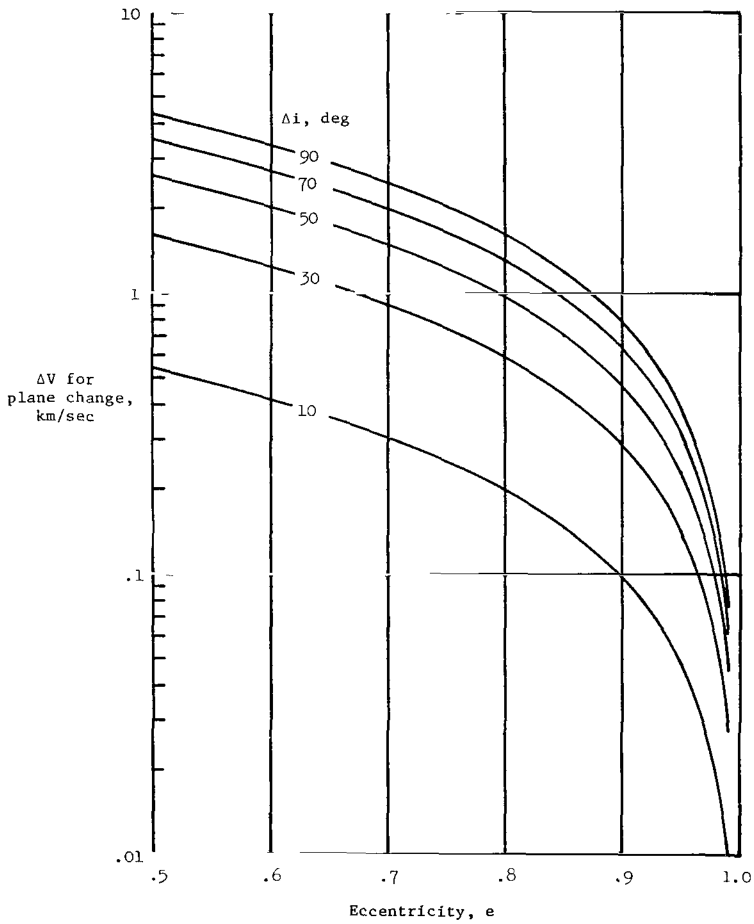


Figure 8. - Requirements of ΔV for plane changes made at the apoapsis of an ellipse for which $r_{PH} = 6878$ km and $0.5 \leqq e < 1.0$.

Assessment of ΔV Penalties Associated With Out-of-Plane Maneuvers

Up to this point it has been assumed that the OOS rendezvous module and the returning spacecraft will occupy coplanar trajectories. Certainly this should be part of the mission plan for use of hyperbolic rendezvous as a primary return mode. However, in an emergency situation a plane change might be required. Figure 8 shows the extra velocity increment which would be required for plane changes as a function of transfer-ellipse eccentricity if it is assumed that the engine burn is made at the apoapsis of the transfer ellipse. Of course, for an infinitely large ellipse, the ΔV for plane change approaches zero. For the 50-hour transfer ellipse previously taken as nominal, a plane change of as much as 90° can be made with a ΔV of only a few hundred meters per second.

Effect of Finite Thrusting Times on the Hyperbolic-Rendezvous Maneuver

Comparison of the M_0 computed for interplanetary return missions of varying V_∞, as presented in figure 7, with the thrust levels previously assumed for the chemical and nuclear propulsion systems shows that low thrust-to-weight ratios can exist. The long resulting thrusting times cause the ΔV requirements to become larger than those obtained in the impulsive approximation. This effect, caused by thrusting against the gravitational field, is commonly known as gravity loss. Thrusting times of both chemical and nuclear systems are given in figure 9 for typical symmetrical rendezvous and deboost maneuvers. For these calculations, constant thrust controlled by a tangential steering law was assumed. Table 4 shows, for nuclear and chemical systems with expendable tanks, the mass of the rendezvous vehicle prior to each of the four burns required for the complete symmetric rendezvous and return mission. Impulsive thrust and the 4500-kg rendezvous module are assumed. The masses in columns 3 and 4 include the 900-kg payload assumed to be transferred from the interplanetary spacecraft.

As an example of the effects of finite thrusting times, for $V_\infty = 6.10$ km/sec (20 000 ft/sec), nominal impulsive requirements for ΔV for departure from circular Earth orbit and the rendezvous maneuver are 2.88 and 4.30 km/sec, respectively. (See table 2.) For this case, a chemical system with expendable tanks has a mass of about 237 000 kg (see fig. 7 or table 4(b)), which results in an initial thrust-to-weight ratio of approximately 0.4 for the engine thrust previously assumed. From figure 9, the first burn time is seen to be about 9 minutes. The gravity loss is 1.3 percent; therefore, ΔV for this example is 0.04 km/sec higher than that required in the impulsive case. For the second burn, the rendezvous maneuver, the thrust-to-weight ratio is up to 0.75;

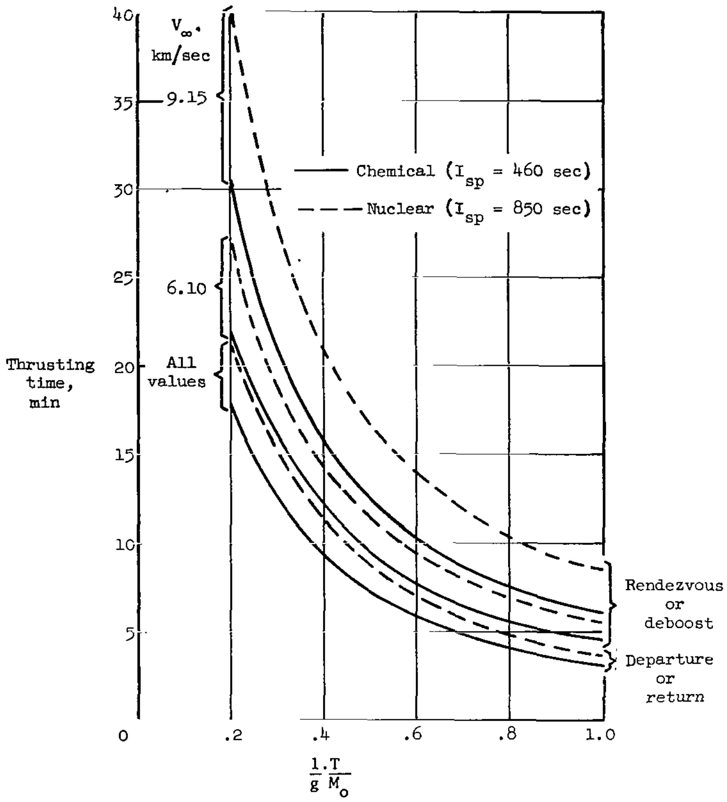


Figure 9.- The effect of thrust-to-weight ratio on thrusting time.

TABLE 4.- SYSTEM MASS PRIOR TO START OF EACH MISSION PHASE

(a) Nuclear propulsion with expendable tanks, 4500-kg rendezvous module

V_∞, km/sec	Total system mass, kg, for[a] –			
	1	2	3	4
1.52	55 810	36 988	32 156	28 745
3.05	67 651	44 836	35 449	28 745
6.10	147 850	97 987	52 620	28 745
9.14	433 567	287 346	90 431	28 745

(b) Chemical propulsion with expendable tanks, 4500-kg rendezvous module

V_∞, km/sec	Total system mass, kg, for[a] –			
	1	2	3	4
1.52	46 123	23 108	18 244	15 119
3.05	63 924	32 026	21 556	15 119
6.10	237 275	118 876	41 944	15 119
9.14	1 451 740	727 327	104 413	15 119

[a]Numbers denote the following maneuvers:
1 departure from 500-km circular Earth orbit
2 rendezvous with return vehicle
3 deboost from hyperbola
4 enter 500-km circular Earth orbit.

this results in a burn time of 6.4 minutes and a gravity loss of 5.3 percent, or an additional ΔV of about 0.23 km/sec. Gravity losses for the third and fourth burns are negligible because of the relatively high thrust-to-weight ratios. The finite burn times and gravity-loss terms have been computed by numerically integrating the trajectories with a computer program similar to that developed in reference 14. To first order, that is, assuming that the thrust-to-weight ratios for each burn remain the same, the gravity losses for the chemical system require an additional mass in Earth orbit of 17 000 kg – an increase of about 7.2 percent over the impulsive case. Such a small penalty does not have a significant effect on any of the conclusions of this study.

CONCLUSIONS

Hyperbolic rendezvous with returning interplanetary spacecraft has been shown to be feasible for a primary or rescue Earth-return mode if the availability of orbit-to-orbit shuttles with masses of several hundred thousand kilograms (initial mass in Earth orbit) is assumed. Both chemical and nuclear systems have possible applications, although the chemical systems are restricted to missions involving small hyperbolic excess velocities. The initial masses in Earth orbit required for the hyperbolic rendezvous maneuver have been shown to compare favorably with those required for transporting an Earth-return retrobraking capability to Mars and back. The Mars missions are considered to be representative of other planetary missions as well.

The usefulness of hyperbolic rendezvous in a rescue operation is enhanced by the demonstrated lack of sensitivity of the required velocity increments to location of the hyperbolic periapsis. This means that a disabled spacecraft can be tracked and recovered successfully with a minimum of maneuvering on its part. Even out-of-plane maneuvers can be accommodated, as shown, for modest weight penalties when time is allotted for sufficiently large transfer ellipses.

Based on the assumptions of weight and engine thrust made in this study, gravity losses due to finite thrust times have been examined. They amount to only a few percent and do not have a significant effect on any of the conclusions of this study.

Langley Research Center,
National Aeronautics and Space Administration,
Hampton, Va., June 24, 1971.

APPENDIX A

TRANSFER ELLIPSES FOR A HYPERBOLIC-RENDEZVOUS MANEUVER

For the geometry shown in figure 2, assume that the hyperbolic true anomaly θ_H associated with a rendezvous point P' is known; that is, either θ_H has been used to specify P', or the angle associated with some other desired quantity – for example, a particular time from hyperbolic periapsis – is known. Clearly, for a particular r_{PE} there will be an allowable range of values for the angular orientation of the transfer ellipse, measured between the elliptic periapis and the hyperbolic axis (Θ in fig. 2). The elliptic true anomaly θ_E is related to Θ and θ_H by

$$\theta_E = \Theta + \theta_H \tag{A1}$$

The angle θ_E and eccentricity e_E are interrelated through the radius r and the elliptic periapsis distance r_{PE}. (All necessary conic relations for deriving the equations in this appendix can be found in ref. 9.) Thus,

$$\cos\theta_E = \frac{r_{PE}\left(1 + e_E\right) - r}{e_E r} \tag{A2}$$

$$e_E = \frac{r - r_{PE}}{r_{PE} - r\cos\theta_E} \tag{A3}$$

The smallest eccentricity will occur for $\theta_E = 180^{\circ}$, which gives

$$e_{E,min} = \frac{r - r_{PE}}{r + r_{PE}} \tag{A4}$$

The limiting largest ellipse, $e_E \rightarrow 1$, occurs for

$$\cos\theta_E \rightarrow \frac{2r_{PE} - r}{r} \tag{A5}$$

A case of special interest occurs when the hyperbola and the ellipse are tangential at P'. For all points on an ellipse,

$$p_E = \frac{r^2 V_E^2 \cos^2\gamma_E}{\mu} \tag{A6}$$

If the ellipse and the hyperbola are tangential at P', then $\gamma_E = \gamma_H$, and since $\gamma_E = 0$ at $r = r_{PE}$,

$$p_E = \frac{r^2 V_E{}^2 \cos^2\gamma_H}{\mu} = \frac{r_{PE}^2 V_{PE}^2}{\mu} \tag{A7}$$

By substituting for V_E, equation (A7) can be rewritten as

$$r^2\left(\frac{2}{r} - \frac{1}{a_E}\right)\cos^2\gamma_H = r_{PE}^2\left(\frac{2}{r_{PE}} - \frac{1}{a_E}\right) \tag{A8}$$

Now a_E can be solved for in terms of known quantities for the tangential ellipse:

$$a_{E,t} = \frac{r^2\cos^2\gamma_H - r_{PE}^2}{2\left(r\cos^2\gamma_H - r_{PE}\right)} \tag{A9}$$

For each set of specified input conditions for r_{PE}, r_{PH}, and V_∞, there exists a value of r beyond which no tangential ellipse can reach without escape velocity being exceeded. At this value of r, a limiting ellipse (parabola) of eccentricity $e_E = 1$ will be tangential to the hyperbola. At $r = r_{max,t}$,

$$\cos^2\gamma_E = \frac{p_E\mu}{r^2 V_E{}^2} \tag{A10}$$

and

$$\cos^2\gamma_H = \frac{p_H\mu}{r^2 V_H{}^2} \tag{A11}$$

Dividing equation (A10) into equation (A11) gives, since $\gamma_E = \gamma_H$,

$$1 = \frac{p_H}{p_E}\frac{V_E{}^2}{V_H{}^2} = \frac{p_H}{p_E}\frac{\dfrac{2}{r_{max,t}}}{\dfrac{2}{r_{max,t}} + \dfrac{1}{a_H}} \tag{A12}$$

Solving for $r_{max,t}$ gives

$$r_{max,t} = \frac{a_H\left(p_H - 2r_{PE}\right)}{r_{PE}} \tag{A13}$$

since $p_E = 2r_{PE}$ when $e_E = 1$.

APPENDIX B

PROPELLANT REQUIREMENTS FOR MULTIBURN MISSIONS WITH AND WITHOUT TANK STAGING

The missions considered for hyperbolic rendezvous require considerable amounts of propellant because of the large ΔV requirements. Therefore, the staging of propellant tanks after each burn can have a significant effect on the total propellant requirement. Propellant requirements can be computed for impulsive burns from the basic rocket equation

$$A = \exp\left(\frac{\Delta V}{I_{sp}g}\right) = \frac{M_{initial}}{M_{final}} \tag{B1}$$

When more than one burn is required, the equation for the nth burn is

$$A_n = \frac{M_{total,n}}{M_{total,n+1}} \tag{B2}$$

where $M_{total,n+1}$ is the total remaining mass at the end of the nth burn. In the case for which tanks are <u>not</u> dropped off after each burn, the equation for the nth burn is

$$A_n = \frac{M_n + \sum_{i=n}^{last} m_i + \alpha \sum_{i=1}^{last} m_i}{M_n + \sum_{i=n+1}^{last} m_i + \alpha \sum_{i=1}^{last} m_i} \tag{B3}$$

The quantity M_n is the total mass of the stage except for propellant and tanks. The total propellant for n burns can be shown, with some manipulation, to be

$$m_{total} = \frac{\sum_{i=1}^{last} M_i\left(\prod_{j=0}^{i-1} A_j\right)\left(A_i - 1\right)}{1 + \alpha\left(1 - \prod_{j=1}^{last} A_j\right)} \tag{B4}$$

if A_0 is defined as being equal to 1. As an example, the propellant required for a three-burn maneuver is

$$m\big|_{n=3} = \frac{M_1\left(A_1 - 1\right) + M_2A_1\left(A_2 - 1\right) + M_3A_1A_2\left(A_3 - 1\right)}{1 + \alpha\left(1 - A_1A_2A_3\right)}$$

The equation, comparable with equation (B3), for the nth burn when tanks are dropped off after the end of each burn is

$$A_n = \frac{M_n + \sum_{i=n}^{last} m_i + \alpha \sum_{i=n}^{last} m_i}{M_n + \sum_{i=n+1}^{last} m_i + \alpha \sum_{i=n}^{last} m_i} \tag{B5}$$

From this equation it can be shown that

$$m_n = \frac{\left[M_n + (1 + \alpha) \sum_{i=n+1}^{last} m_i\right]\left(A_n - 1\right)}{\left[1 + \alpha\left(A - A_n\right)\right]} \tag{B6}$$

which gives the propellant requirement for the nth burn in terms of the sum of the propellants for the succeeding burns. The propellant for the last burn is

$$m_{last} = \frac{M_{last}\left(A_{last} - 1\right)}{\left[1 + \alpha\left(1 - A_{last}\right)\right]} \tag{B7}$$

Now, given a particular mission, the total propellant can easily be computed by starting from the last burn and working backwards.

APPENDIX C

MASS REQUIREMENTS FOR MANNED MARS MISSIONS

The purpose of this appendix is to document the procedures used for generating the data shown in figure 7 for the manned Mars missions. Three cases are required: retrobraking, aerobraking, and no Earth-return capability on the spacecraft. The last case is, of course, the hyperbolic-rendezvous case.

The manned Mars missions chosen for comparing conventional Earth-return methods, that is, aerobraking or propulsive braking to Earth orbit, with hyperbolic rendezvous are inbound or outbound Venus swingbys in the period 1982-1999. Eight opportunities are examined. The dates for arrivals and departures are given in table 5 along with the hyperbolic excess velocities, expressed as a fraction of the Earth's mean orbital speed, for each maneuver (ref. 15). Note that the Mars stay times are always 30 days, and the total trip time is not more than 560 days. Clearly, other equally valid criteria could have been used for selecting mission schedules.

TABLE 5.- ARRIVAL AND DEPARTURE DATES AND HYPERBOLIC EXCESS VELOCITIES FOR SELECTED MANNED MARS MISSION OPPORTUNITIES USED IN DETERMINING REPRESENTATIVE INITIAL MASSES REQUIRED IN EARTH ORBIT

Year	Venus swingby	Leave Earth (a)	Arrive Mars (a)	Leave Mars (a)	Arrive Earth (a)	Hyperbolic excess velocities, EMOS,[b] for –			
						Leave Earth	Arrive Mars	Leave Mars	Arrive Earth
1982	Inbound	4970	5190	5220	5530	0.113	0.143	0.219	0.204
84	Inbound	5680	5960	5990	6200	.129	.120	.299	.278
86	Outbound	6150	6530	6560	6670	.140	.172	.181	.203
88	Inbound	7350	7550	7580	7870	.115	.089	.240	.258
90	Outbound	7830	8160	8190	8390	.140	.184	.282	.214
93	Outbound	8510	8820	8850	9070	.159	.208	.125	.141
95	Inbound	9660	9890	9920	0180	.126	.142	.214	.283
99	Outbound	0840	1180	1210	1360	.159	.191	.195	.198

[a]Julian dates, 244____ or 245____.

[b]Earth mean orbital speed about the Sun, 29.78 km/sec.

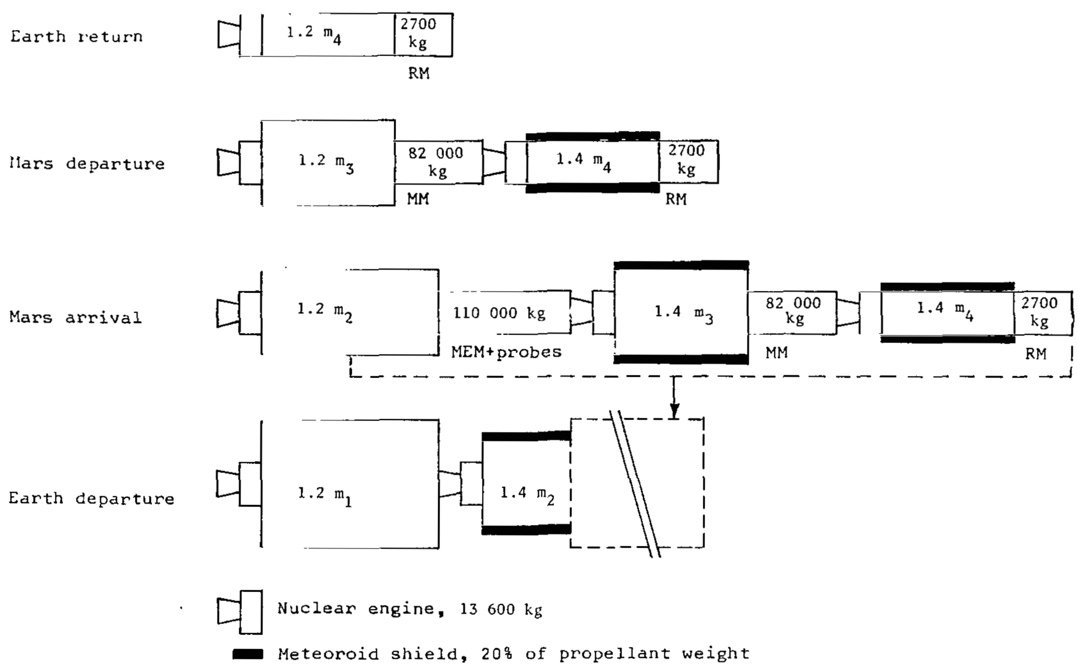


Figure 10.- Schematic representation of weights associated with each stage of a Manned Mars mission just prior to the major rocket burns associated with the mission.

There are many operational details and component masses which need to be specified for the complete definition of a mission. For the purposes of this study the following mission sequence and specifications are followed for a retrobraked Earth return:

(1) The initial mass in Earth orbit M_0 just prior to departure consists of a 50 000-kg Mars excursion module (MEM); a 37 200-kg mission module (MM); a 2700-kg Earth retromodule (RM); four 13 600-kg nuclear engines (one for each burn), propellant for each of the four burns; tanks and propellant reserve for each burn which, taken together, weigh 20 percent of the respective propellant load; and meteoroid shields (an additional 20 percent of propellant) for each burn except the first.

(2) Launch from a 185-km circular Earth orbit and jettison engine.

(3) Jettison meteoroid shield just prior to Mars orbit insertion.

(4) Deboost into very low circular Mars orbit and jettison engine.

(5) Jettison meteoroid shield just prior to Mars orbit departure.

(6) Leave Mars orbit and jettison engine.

(7) Jettison meteoroid shield for retrostage just prior to Earth capture.

(8) Deboost into 500-km circular Earth orbit.

From the well-known ideal rocket equation

$$\exp\left(\Delta V/I_{sp}g\right) = \frac{M_{initial}}{M_{final}}$$

the initial mass in Earth orbit can be obtained by working backwards from the final burn. The makeup of each stage just prior to a particular burn is shown schematically in figure 10.

When aerobraking is assumed, an 8200-kg aerobraking capsule replaces the RM, tanks, propellant, and meteoroid shield for that maneuver. Steps 7 and 8 are replaced by the aerobraking maneuver. The M_0 required when hyperbolic rendezvous is used does not include Earth-return capability, as mentioned previously.

REFERENCES

1. Allen, H. Julian; and Eggers, A. J., Jr.: A Study of the Motion and Aerodynamic Heating of Ballistic Missiles Entering the Earth's Atmosphere at High Supersonic Speeds. NACA Rep. 1381, 1958. (Supersedes NACA TN 4047.)
2. Becker, J. V.; Baradell, D. L.; and Pritchard, E. B.: Aerodynamics of Trajectory Control for Re-Entry at Escape Speed. Astronautica Acta, vol. VII, Fasc. 5-6, 1961, pp. 334-358.
3. Chapman, Dean R.: An Analysis of the Corridor and Guidance Requirements for Supercircular Entry Into Planetary Atmospheres. NASA TR R-55, 1960.
4. Love, E. S.; and Pritchard, E. B.: A Look at Manned Entry at Circular to Hyperbolic Velocities. 2nd Manned Space Flight Meeting (Dallas, Texas), Amer. Inst. of Aeron. and Astronaut., Apr. 1963, pp. 167-180.
5. Pritchard, E. Brian: Velocity Requirements and Re-Entry Flight Mechanics for Manned Mars Missions. J. Spacecraft Rockets, vol. 1, no. 6, Nov.-Dec. 1964, pp. 605-610.
6. Staylor, W. Frank; Brooks, David R.; and Pritchard, E. Brian: Requirements and Capabilities of Interorbital Shuttles. J. Spacecraft Rockets, vol. 7, no. 7, July 1970, pp. 814-818.
7. Luidens, Roger W.; Burley, Richard R.; Eisenberg, Joseph D.; Kappraff, Jay M.; Miller, Brent A.; Shovlin, Michael D.; and Willis, Edward A., Jr.: Manned Mars Landing Mission by Means of High-Thrust Rockets. NASA TN D-3181, 1966.
8. Mueller, George E.: An Integrated Space Program for the Next Generation. Astronaut. Aeronaut., vol. 8, no. 1, Jan. 1970, pp. 30-51.
9. Anon.: Basic Techniques and Data. Pt. 1 of Orbital Flight Handbook. NASA SP-33, Pt. 1, 1963.
10. Eggers, Alfred J., Jr.; and Swenson, Byron L.: Lifting Entry Vehicles for Future Space Missions. Astronaut. Aeronaut., vol. 5, no. 3, Mar. 1967, pp. 24-33.
11. Corrington, L. C.: The Nuclear Rocket Program – Its Status and Plans. J. Spacecraft Rockets, vol. 6, no. 4, Apr. 1969, pp. 465-470.
12. Dugan, Duane W.: The Role of Staged Space Propulsion Systems in Interplanetary Missions. NASA TN D-5593, 1969.
13. Mission Evaluation Team: Apollo 11 Mission Report. MSC-00171, NASA, 1969. (Available as NASA TM X-62633.)

14. Williams, James L.; and Barker, L. Keith: Analytical Study of Lunar Landing Trajectories With Reference to the Lunar-Orbit-Rendezvous Mode and Possible Abort Situations. NASA TN D-2016, 1963.

15. Douglas Aircraft Co.: Mars Stopover Missions Using Venus Swingbys. Pt. 6 of Planetary Flight Handbook. NASA SP-35, Pt. 6, 1968.

NASA-Langley, 1971 —— 31 L-7664

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
WASHINGTON, D. C. 20546

OFFICIAL BUSINESS
PENALTY FOR PRIVATE USE $300

FIRST CLASS MAIL

POSTAGE AND FEES PAID
NATIONAL AERONAUTICS AND
SPACE ADMINISTRATION

013 001 C1 U 31 710903 S00903DS
DEPT OF THE AIR FORCE
AF SYSTEMS COMMAND
AF WEAPONS LAB (WLOL)
ATTN: E LOU BOWMAN, CHIEF TECH LIBRARY
KIRTLAND AFB NM 87117

POSTMASTER: If Undeliverable (Section 158 Postal Manual) Do Not Return

"The aeronautical and space activities of the United States shall be conducted so as to contribute . . . to the expansion of human knowledge of phenomena in the atmosphere and space. The Administration shall provide for the widest practicable and appropriate dissemination of information concerning its activities and the results thereof."

—NATIONAL AERONAUTICS AND SPACE ACT OF 1958

NASA SCIENTIFIC AND TECHNICAL PUBLICATIONS

TECHNICAL REPORTS: Scientific and technical information considered important, complete, and a lasting contribution to existing knowledge.

TECHNICAL NOTES: Information less broad in scope but nevertheless of importance as a contribution to existing knowledge.

TECHNICAL MEMORANDUMS: Information receiving limited distribution because of preliminary data, security classification, or other reasons.

CONTRACTOR REPORTS: Scientific and technical information generated under a NASA contract or grant and considered an important contribution to existing knowledge.

TECHNICAL TRANSLATIONS: Information published in a foreign language considered to merit NASA distribution in English.

SPECIAL PUBLICATIONS: Information derived from or of value to NASA activities. Publications include conference proceedings, monographs, data compilations, handbooks, sourcebooks, and special bibliographies.

TECHNOLOGY UTILIZATION PUBLICATIONS: Information on technology used by NASA that may be of particular interest in commercial and other non-aerospace applications. Publications include Tech Briefs, Technology Utilization Reports and Technology Surveys.

Details on the availability of these publications may be obtained from:

SCIENTIFIC AND TECHNICAL INFORMATION OFFICE

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

Washington, D.C. 20546